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ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

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AGARD ADVISORY REPORT No.167

AGARD Three-Dimensional Aeroelastic Configurations

NORTH ATLANTIC TREATY ORGANIZATION

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NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARD Advisory Report No.167

AGARD THREE-DIMENSIONAL AEROELASTIC CONFIGURATIONS

Compiled by

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PREFACE

At its Fall 1977 meeting in Voss, Norway the AGARD Structures and Materials Panel (SMP) formed a Working Group on "Standard Configurations for Aeroelastic Applications of Transonic Unsteady Aerodynamics". The members were:

S.R.Bland, United States (Coordinator)
F.O.Carta, United States
L.Chesta, Italy
R.Dat, France
H.Försching, Federal Republic of Germany (Deputy Chairman)
H.C.Garner, United Kingdom
W.Geissler, Federal Republic of Germany
J.J.Olsen, United States (Chairman)
J.J.Philippe, France (Fluid Dynamics Panel Representative)
H.Tijdeman, Netherlands
J.C.Uselton, United States (Fluid Dynamics Panel Representative)

The aim of the Working Group was to accelerate the development of new theoretical, numerical and experimental techniques in transonic unsteady aerodynamics and their application to aeroelastic problems of aircraft loads, stability and flutter. The members from six nations obtained numerous suggestions from aeroelasticians and aerodynamicists in their countries and worked diligently to mold the recommendations into a number which was manageable, yet constituted a valid test of newly emerging capabilities. Their first product was a standard set of two dimensional airfoils and aerodynamic conditions, contained in AGARD-AR-156 "AGARD Two Dimensional Aeroelastic Configurations." This report constitutes the second product of the Working Group, a standard set of three dimensional wings and aerodynamic conditions.

JAMES J.OLSEN
Chairman, Working Group on
Standard Aeroelastic Configurations

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AGARD THREE-DIMENSIONAL AEROELASTIC CONFIGURATIONS

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SUMMARY

The aeroelastician needs reliable and efficient methods for the calculation of unsteady aerodynamic forces in the frequently critical transonic speed regime. The development of such methods may be enhanced by the availability of a limited number of test cases for the comparison of competing methods. This report contains such test cases for five clean, isolated wings. Wing geometric descriptions, airfoil coordinates, and suggested aerodynamic conditions for each are included.

LIST OF SYMBOLS

AR	full-span aspect ratio, $span^2/area$
c	local streamwise chord, m
c_r	root chord, m
f	oscillation frequency, Hz
h	plunge displacement in z-direction, m
h_o	plunge amplitude, m
k	reduced frequency, $\omega c_r/2V$
M	free-stream Mach number
p	static pressure, N/m^2
Re	Reynolds number, Vc_r/ν
s	semispan, m
S	area of planform, m^2
t	time, s
x	streamwise coordinate relative to root leading edge, positive downstream, m
x_α	pitch axis relative to local leading edge, m
x_δ	control hinge relative to local leading edge, m
y	spanwise coordinate relative to root, positive to right looking upstream, m
z	vertical coordinate, positive up, m
α	streamwise angle of attack, deg
α_m	mean α, deg
α_o	dynamic pitch angle in streamwise direction, deg
δ	streamwise control deflection angle, deg
δ_m	mean δ, deg
δ_o	dynamic control angle in streamwise direction, deg
η	nondimensional spanwise location, y/s
Λ	sweep angle, positive for sweep back, deg
ν	kinematic viscosity, m^2/s
ξ	streamwise distance from local leading edge as fraction of local chord
ρ	free-stream density, kg/m^3
ω	angular frequency, $2\pi f$, rad/s

The coordinate system, force and moment definitions, and sign conventions are given in figure 1.

1. INTRODUCTION

The technology of transonic aerodynamics is currently undergoing rapid development. Significant progress is being made in the solution of the equations describing the unsteady motion of airfoils and wings in transonic flow. The availability of reliable and efficient computational methods will greatly enhance the ability of the analyst to predict the aeroelastic behavior of high-speed aircraft. In general, solution of the equations for the transonic regime, with their inherent nonlinearity in the time-dependent displacements, requires the use of the finite-difference or finite-element methods of the computational fluid dynamicist. These methods tend to be expensive to use, requiring both large computer storage and long machine time. The aeroelastician needs to examine many cases, both for analysis and for structural design optimization, and therefore is interested in the development of reliable, more approximate methods.

In order to compare and evaluate analytical methods involving various degrees of approximation, the AGARD Structures and Materials Panel has defined a limited set of test cases to be used in evaluating the competing methods. This activity should serve to stimulate cooperative research and to conserve resources by providing a common set of analytical problems.

Recommended test cases for two-dimensional airfoils have been published in reference 1. This report contains the recommended cases for five isolated clean wings.

The wings are: (1) a rectangular wing of aspect ratio four with symmetric airfoil section; (2) a swept, tapered wing of aspect ratio six with symmetric section and a control surface; (3) a low aspect ratio thin, cambered wing; (4) a high aspect ratio, transport type wing with supercritical airfoil section and a control surface; and (5) a high aspect ratio wing with supercritical airfoil section. Detailed geometric definitions of the wing planforms and airfoil sections are given. In addition, the aerodynamic conditions such as Mach number, mean angle of attack, and oscillation mode, amplitude, and frequency are also included. Experimental data for some of the cases are or will be available for comparison. No data are included herein concerning test conditions such as static and elastic deformation, natural vs. fixed transition, size of wind tunnel test section, or tunnel wall conditions, for which reference may be made to the test reports. Recommendations are made for uniformity in definitions and reporting to enhance the desired comparisons of analytical methods.

2. WING GEOMETRY

Sketches of the five AGARD semispan planforms are given in figure 2. The axes for pitch or control-surface rotation are shown by dashed lines. Tabulated airfoil ordinates for each wing are given in tables 1-5. Because of the sensitivity of transonic calculations to surface slopes (and curvature for some methods), care should be taken to ensure that interpolations of the geometric data are as smooth as possible. The use of low-order least-square polynomials or spline functions is recommended for interpolation in the chordwise (streamwise) direction. Linear interpolation along constant percent chord lines is to be used in the spanwise direction. Whatever geometric description is actually used in the aerodynamic analysis, it should be carefully documented so that the calculation conditions can be duplicated by other analysts. Detailed descriptions of the five wings are given in the following subparagraphs. In each case, planform dimensions are in terms of unit root chord. In figures 3-5 and 7-8 the x-y coordinates of the points needed to define the planform geometry are given.

2.1 Rectangular Wing

This unswept, rectangular wing, which has a full-span aspect ratio of four, is shown in figure 3. A model with chord of 0.2 m will be built and tested by the RAE. The airfoil ordinates, given in table 1, are a symmetric version of the NACA 64A010A tested as a two-dimensional section at the NASA Ames Research Center. The actual thickness-to-chord ratio is about 10.6%. This airfoil is one of the AGARD two-dimensional standard configurations (ref. 1). The pitch axis has two locations, at quarter-chord and mid-chord, both of which are in the experimental program.

2.2 RAE Wing A

The RAE Wing A, shown in figure 4, has full-span aspect ratio of six, taper ratio of one-third, and midchord sweep-back angle of 30°. The airfoil is the symmetric 9% RAE 101 section with streamwise ordinates given in table 2. This airfoil has nose radius of 0.006184c. The control surface is of 30% local chord and extends from 40% to 70% semispan. The control-surface oscillation axis is at the control leading edge. The unswept pitch axis is at mid-root chord. Some experimental results for a wind-tunnel model with oscillating control surface and root chord of 0.24 m are given in reference 2. The wing, without control surface, is an AGARD Fluid Dynamics Panel standard (ref. 3) for steady flow calculations.

2.3 NORA Model

The NORA model is a model of the Mirage F-1 horizontal tail which has been extensively tested in four European wind tunnels (ref. 4). The acronym NORA refers to NLR, ONERA, RAE, and AVA. The wind tunnel model has a root chord of 0.65 m and a non-streamwise tip. The analytical planform, shown in

figure 5, has been chosen to have a streamwise tip; all other geometry is identical to that of the experimental model. The NORA planform has a leading edge sweep angle of 50° and has approximately aspect ratio of 2.01, taper ratio of 0.35, and trailing edge sweep back angle of 13.45°. The pitch axis is swept back 35° and intersects the root chord at about $x/c_r = 0.526$.

The airfoil sections are based on the symmetric NACA 63006 profile modified to a thickness ratio of about 5% and with a small updroop near the nose. The wing is defined by data at three sections: Ⓐ the root chord, Ⓑ η about 0.28, and Ⓒ η about 1.06. The nose camber is given in figure 6. The mean camber line for each section is defined in terms of the offset ζ from the z-plane at 5 equispaced (Δ) points. The region of cambering covers the leading 8.15% chord at the root and the leading 15.98% chord at section Ⓒ ; the region of cambering varies linearly in the spanwise direction, as shown in figure 5. The airfoil ordinates are given in table 3. In the cambered region, the ordinates give the increment to be added to the camber line, in a direction perpendicular to the camber line, to define the upper surface (subtracted for lower surface). Spanwise linear interpolation along constant percent local chord lines is used to define the wing surface at points between the defining sections.

2.4 ZKP Wing

The ZKP wing is a model of a transport-type wing designed by VFW and tested at ONERA. The model has aspect ratio 8.84, root chord of about 1.802m, and leading edge sweep of about 30°. This planform is shown in figure 7. The supercritical airfoil shape is definedd at three stations, η = 0.15, 0.4, and 0.85, by the ordinates given in table 4. Linear interpolation (or extrapolation) along constant percent chord lines is used to define the wing surface at other stations. The wing root has positive incidence. The twist defined by the ordinates produces negative incidence outboard. The control surface hinge line lies at 77.4% local chord and its side edges are at η = 0.8389 and 0.9896.

2.5 LANN Wing

The LANN Wing is a supercritical research wing model, built by the Lockheed-Georgia Company for the U.S. Air Force for testing at NLR and NASA Langley. The model has a span of one meter, a quarter-chord sweep angle of 25° and an aspect ratio of about 7.9. The planform is shown in figure 8. The unswept pitch axis is at 62.1% root chord. The supercritical airfoil shape is defined at $\eta = 0$ and $\eta = 1$ by the ordinates given in table 5. Linear interpolation along constant percent chord lines defines the wing surface at other stations. The airfoil thickness is about 12% and the wing is twisted from about 2.6° at the root to about -2.0° at the tip.

3. ANALYTICAL TEST CASES

The suggested analytical test cases for the three wings are given in tables 6-10. The reduced frequency k uses the root semichord $c_r/2$ as reference length. An attempt has been made to cover a range of conditions for each wing while at the same time limiting the total number of cases. Of the 46 cases listed, there is a subset of 16 cases, which have been chosen for priority analysis and are indicated by asterisks in the tables. These cases provide for the systematic variation of one parameter at a time. It is recommended that calculations of each mean steady flow ($\alpha = \alpha_m$, $\delta = \delta_m$) condition be made.

The modes of motion are described as follows. For wing pitch about a mean angle of attack,

$$\alpha(t) = \alpha_m + \alpha_0 \sin \omega t$$

For the plunge mode,

$$h(t) = h_0 \sin \omega t$$

For the control surface mode,

$$\delta(t) - \delta_m + \delta_0 \sin \omega t$$

In each case, angles are specified in the streamwise direction.

3.1 Rectangular Wing

The analytical cases for the rectangular wing are given in table 6. All cases involve wing pitch oscillation about a mean angle of zero. At M = 0.8, frequency, amplitude, and Reynolds number are varied. These cases were selected to correspond with the two-dimensional ones in table 8 of reference 1. There is one case at M = 0.9. This wing provides a good test vehicle for comparing two- and three-dimensional results and for investigating the suitability of strip theory. It is recommended that pressure be calculated at stations $\eta = \cos(n\pi/9)$ for n = 1, 2, 3, 4, at which measurements will be made in the RAE tests.

3.2 RAE Wing A

The analytical cases for the RAE Wing A are presented in table 7. These cases include pitch, plunge, and control-surface oscillation. Given are variations in amplitude, mean angle, and frequency. The quasi-steady $k = 0.003$ is included. Experimental results for case 4 are given in reference 2; additional experimental data are available for cases 5 and 8-11. The experimental measurement stations are $\eta = 0.35$, 0.45, 0.60, and 0.75.

3.3 NORA Model

The analytical cases for the NORA model are given in table 8. These cases were selected from those in the experimental program (ref. 4) and involve pitch oscillation about the swept back axis in figure 5 at four Mach numbers, including the supersonic $M = 1.1$. Mean angle of attack and frequency are varied. The experimental measurement stations are at $\eta = 0.524$ and 0.712.

3.4 ZKP Wing

The analytical cases for the ZKP Wing are given in table 9. Only the control surface mode of oscillation is included. One case involves a nonzero angle-of-attack; several mean flap angles and Mach numbers are listed. In the experiments the slot between the wing and the control surface leading edge was sealed. The experimental unsteady pressures were measured at $\eta = 0.405$, 0.640, and 0.885. A comparison between calculated and measured data for several of these cases appears in reference 5. The wing design condition is at $M = 0.78$, $\alpha_m = 1.5^o$, and lift coefficient 0.5.

3.5 LANN Wing

The test cases for the LANN wing are given in table 10. The single mode of oscillation is pitch about the 62.1% chord axis. Several Mach numbers, amplitudes, and frequencies are included with the priority cases chosen at the design condition, $M = 0.82$. In addition to experimental results at moderate Reynolds numbers, it is anticipated that tunnel tests will be conducted at flight Reynolds numbers. The pressure measurement stations lie at $\eta = 0.200$, 0.325, 0.475, 0.650, 0.825, and 0.950.

4. RECOMMENDATIONS FOR REPORTING RESULTS

Although it is impossible to require a single uniform format for reporting results obtained from different analytical methods, as much uniformity as is practical will certainly enhance the comparisons between various investigations that this AGARD activity is designed to promote. In any case, we again urge that such details as sign conventions, units and nondimensionalizing factors be clearly reported.

In addition to the pressure distributions at several span stations, section and total force and moment coefficients should be reported. The recommended definitions and sign conventions for these are shown in figure 1. In comparing results from either different methods or from the same nonlinear method at different amplitudes, it is desirable to nondimensionalize further by dividing the pressure coefficient, say, by the amplitude. In this case, a symbol such as C_p/α_0 should be used for the pressure coefficient per radian.

In unsteady aerodynamics the coefficients are, of course, functions of time. The aeroelastician has traditionally worked with complex coefficients for harmonic motion. These may be expressed as real (in-phase with the motion) and imaginary (in-quadrature) parts, or alternatively, as magnitude and phase. For nonlinear aerodynamics, this representation is inadequate. In general, the coefficients are computed as functions of time. A Fourier analysis can be made and higher harmonics reported along with the fundamental. In many cases a spectral analysis may be more appropriate. In addition to pressure and force coefficients, the shock wave strength, amplitude, and phase with respect to the motion are important. A comparison of the mean values of all the unsteady flow parameters with the corresponding parameters for the mean steady flow is also of interest.

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Table 1.- Ordinates for Rectangular Wing
Upper surface (symmetric airfoil)

ξ	z/c	ξ	z/c	ξ	z/c	ξ	z/c
.00000	.00000	.01798	.01579	.09002	.03217	.44999	.05232
.00102	.00380	.02200	.01727	.10003	.03372	.50003	.05010
.00198	.00532	.02601	.01860	.10998	.03516	.55001	.04693
.00300	.00678	.03002	.01981	.11999	.03654	.60000	.04301
.00401	.00789	.03399	.02091	.13000	.03785	.64999	.03847
.00498	.00876	.03800	.02196	.14001	.03907	.70003	.03351
.00605	.00959	.04201	.02294	.15001	.04024	.75001	.02820
.00701	.01029	.04602	.02384	.20000	.04514	.80000	.02278
.00798	.01091	.04999	.02471	.24999	.04886	.84999	.01730
.00899	.01154	.05999	.02679	.30003	.05144	.90003	.01176
.01001	.01211	.07000	.02871	.35001	.05295	.95001	.00623
.01402	.01412	.08001	.03047	.40508	.05314	1.00000	.00000

Table 2.- Ordinates for RAE Wing A
Upper surface (symmetric airfoil)

ξ	z/c	ξ	z/c	ξ	z/c	ξ	z/c
.000000	.000000	.182803	.040576	.549008	.035330	.886505	.009134
.002408	.005448	.222215	.042852	.597545	.031957	.915735	.006782
.009607	.010822	.264302	.044387	.645142	.028410	.940961	.004752
.021530	.016056	.308658	.044994	.691342	.024813	.961940	.003064
.038060	.021079	.354858	.044487	.735698	.021263	.978470	.001733
.059039	.025819	.402455	.043116	.777785	.017885	.990393	.000773
.084265	.030208	.450992	.041044	.817197	.014713	.997592	.000194
.113495	.034178	.500000	.038403	.853553	.011787	1.000000	.000000
.146447	.037658						

Table 3. - Airfoil coordinates for NORA Model at the defining sections.

ξ	z/c Ⓐ	z/c Ⓑ	z/c Ⓒ
0	0	0	0
.0050	.005731	.005468	.006023
.0125	.008308	.007839	.007990
.025	.010654	.009998	.009834
.050	.013269	.012345	.011309
.10	.018011	.016504	.014136
.15	.021380	.019591	.016732
.20	.023800	.021808	.018625
.25	.025457	.023327	.019923
.30	.026469	.024255	.020715
.35	.026882	.024632	.021039
.40	.026622	.024394	.020833
.45	.025780	.023622	.020174
.50	.024400	.022359	.019092
.55	.022554	.020667	.017652
.60	.020314	.018613	.015896
.65	.017760	.016273	.013890
.70	.014965	.013712	.011712
.75	.012025	.011018	.009411
.80	.009032	.008277	.007066
.85	.006120	.005607	.004789
.90	.003431	.003139	.002685
.95	.001235	.001132	.000964
1.00	0	0	0

Ordinates measured perpendicular to mean camber line (defined in fig. 6).

Ordinates measured from z = 0 plane.

Table 4.- Ordinates for ZKP Wing

$\eta = 0.15$

ξ	z/c	ξ	z/c	ξ	z/c	ξ	z/c
.999777	-.083386	.243032	-.103337	.000909	.010701	.305386	.044828
.974507	-.084610	.218141	-.099889	.001349	.012321	.330387	.042741
.949231	-.085903	.193301	-.095811	.001701	.013401	.355360	.040294
.923949	-.087277	.173472	-.091999	.002009	.014241	.380310	.037566
.898661	-.088720	.153683	-.087716	.003155	.016751	.405231	.034488
.873369	-.090214	.133944	-.082814	.004273	.018670	.430127	.031090
.848069	-.091808	.114259	-.077240	.005369	.020239	.455001	.027433
.822764	-.093451	.094636	-.070897	.006533	.021629	.479854	.023515
.797455	-.095154	.084850	-.067425	.008607	.023707	.504687	.019337
.772143	-.096899	.075079	-.063754	.010779	.025505	.529500	.014929
.746829	-.098662	.065331	-.059792	.014027	.027683	.554297	.010311
.721520	-.100365	.055611	-.055501	.017191	.029450	.579081	.005543
.696209	-.102099	.045921	-.050839	.020333	.030957	.603859	.000695
.670900	-.103793	.036265	-.045746	.026566	.033321	.628628	-.004263
.645598	-.105416	.026674	-.039864	.032762	.035236	.653395	-.009241
.620297	-.107020	.020952	-.035917	.043046	.037886	.678165	-.014190
.595001	-.108564	.015297	-.031159	.053288	.040035	.702932	-.019177
.569715	-.109987	.012497	-.028445	.063504	.041855	.727702	-.024126
.544441	-.111261	.009724	-.025391	.073697	.043395	.752471	-.029074
.519182	-.112334	.007020	-.021797	.083874	.044724	.777238	-.034062
.493945	-.113157	.005266	-.018914	.094032	.045844	.802013	-.038940
.468722	-.113791	.003466	-.015251	.104177	.046784	.826780	-.043918
.443522	-.114154	.002592	-.013069	.124428	.048192	.851558	-.048766
.418349	-.114178	.001848	-.010768	.144636	.049071	.876330	-.053695
.393203	-.113881	.001188	-.008306	.164808	.049500	.901104	-.058593
.368086	-.113204	.000520	-.004824	.184955	.049629	.925880	-.063450
.343005	-.112097	.000321	-.003312	.205077	.049457	.950656	-.068329
.317956	-.110599	.000145	-.001517	.230203	.048900	.975434	-.073177
.292942	-.108662	.000019	.000822	.255297	.047943	1.000223	-.077885
.267968	-.106234	.000205	.006460	.280357	.046576		

Table 4.- (Continued)

$\eta = 0.40$

ξ	z/c	ξ	z/c	ξ	z/c	ξ	z/c
.999948	-.012354	.248745	-.059813	.000267	.014803	.301162	.065040
.975004	-.008854	.223765	-.058263	.000593	.016542	.326177	.065290
.950045	-.006244	.198795	-.056212	.000965	.017869	.351188	.065371
.925072	-.004364	.178827	-.054098	.001349	.018981	.376197	.065340
.900085	-.003184	.158867	-.051594	.002442	.021362	.401203	.065120
.875085	-.002654	.138918	-.048590	.003485	.023053	.426206	.064810
.850074	-.002724	.118976	-.045136	.004436	.024324	.451206	.064330
.825051	-.003414	.099044	-.041192	.005393	.025405	.476204	.063679
.800015	-.004814	.089083	-.038970	.007323	.027246	.501198	.062839
.774967	-.006784	.079128	-.036417	.009354	.028828	.526187	.061809
.749905	-.009525	.069174	-.033845	.012389	.030697	.551173	.060579
.724829	-.012925	.059225	-.030993	.015421	.032266	.576154	.059078
.699743	-.016925	.049282	-.027821	.018448	.033598	.601130	.057328
.674650	-.021274	.039346	-.024279	.024495	.035900	.626102	.055328
.649555	-.025714	.029418	-.020347	.030536	.037885	.651068	.053078
.624455	-.030404	.023467	-.017672	.040596	.040763	.676028	.050517
.599358	-.034974	.017525	-.014556	.050646	.043242	.700984	.047728
.574265	-.039323	.014556	-.012798	.060690	.045319	.725932	.044517
.549174	-.043513	.011591	-.010792	.070731	.047247	.750878	.041188
.524089	-.047434	.008639	-.008393	.080768	.048965	.775819	.037557
.499011	-.050973	.006592	-.006395	.090802	.050523	.800755	.033737
.473944	-.054024	.004645	-.004126	.100833	.051991	.825685	.029587
.448885	-.056553	.003681	-.002777	.120891	.054597	.850612	.025267
.423837	-.058564	.002752	-.001257	.140941	.056763	.875530	.020526
.398798	-.060073	.001862	.000562	.160986	.058679	.900445	.015597
.373771	-.060993	.000823	.003581	.181024	.060265	.925354	.010317
.348751	-.061553	.000601	.004504	.201057	.061591	.950257	.004806
.323737	-.061723	.000383	.005606	.226092	.062921	.975156	-.000974
.298733	-.061463	.000182	.007054	.251120	.063911	1.000052	-.006853
.273734	-.060873	.000000	.010570	.276143	.064581		

Table 4.- (Concluded)
$\eta = 0.85$

ξ	z/c	ξ	z/c	ξ	z/c	ξ	z/c
1.000016	.000104	.250339	-.062148	.000231	-.001326	.299635	.061370
.975000	.002851	.225334	-.061490	.000473	.000124	.324633	.062233
.949987	.004809	.200326	-.060342	.000716	.001230	.349629	.062915
.924979	.006026	.180317	-.059017	.000961	.002151	.374626	.063437
.899975	.006464	.160306	-.057320	.001943	.005038	.399626	.063780
.874975	.006240	.140295	-.055320	.003040	.007364	.424626	.063923
.849980	.005348	.120281	-.053000	.003922	.009025	.449628	.063886
.824988	.003795	.100264	-.050323	.004912	.010596	.474630	.063659
.800000	.001551	.090255	-.048831	.006897	.013247	.499633	.063290
.775015	-.001400	.080245	-.047217	.008886	.015465	.524639	.062684
.750035	-.005063	.070236	-.045454	.011870	.018205	.549645	.061846
.725057	-.009286	.060225	-.043491	.014857	.020442	.574650	.060799
.700083	-.014039	.050211	-.041249	.017846	.022319	.599660	.059521
.675112	-.019101	.040194	-.038586	.023829	.025414	.624670	.058034
.650140	-.024184	.030175	-.035313	.029814	.027947	.649679	.056316
.625166	-.029087	.024162	-.032926	.039794	.031495	.674693	.054359
.600191	-.033740	.018146	-.030022	.049778	.034452	.699706	.052231
.575215	-.038112	.015136	-.028328	.059764	.037049	.724720	.049764
.550237	-.042116	.012123	-.026386	.069752	.039276	.749737	.047036
.525257	-.045787	.009111	-.024033	.079741	.041294	.774755	.044010
.500275	-.049021	.007099	-.022164	.089731	.043140	.799776	.040734
.475290	-.051872	.005087	-.019825	.099721	.044787	.824797	.037145
.450302	-.054345	.004079	-.018397	.119706	.047711	.849819	.033397
.425314	-.056450	.003069	-.016710	.139691	.050237	.874844	.029400
.400322	-.058201	.002057	-.014639	.159681	.052410	.899867	.025243
.375330	-.059664	.001041	-.011849	.179672	.054265	.924895	.020805
.350335	-.060856	.000784	-.010927	.199663	.055908	.949922	.016059
.325339	-.061690	.000529	-.009825	.224653	.057642	.974953	.010970
.300341	-.062211	.000271	-.008380	.249647	.059093	.999984	.005603
.275341	-.062364	.000000	-.004854	.274641	.060327		

Table 5.- Ordinates for LANN Wing
$\eta = 0$ (upper surface)

ξ	z/c	ξ	z/c	ξ	z/c	ξ	z/c
.000000	.020816	.017500	.041980	.200000	.058098	.650000	.028709
.000625	.025380	.020000	.043095	.225000	.057863	.700000	.022362
.001250	.027184	.025000	.045044	.250000	.057407	.750000	.015541
.002500	.029683	.035000	.048162	.300000	.055983	.800000	.008356
.003750	.031568	.050000	.051473	.350000	.053944	.850000	.000812
.005000	.033138	.070000	.054230	.400000	.051363	.900000	-.006895
.006250	.034489	.075000	.054718	.450000	.048141	.940000	-.013182
.007500	.035673	.100000	.056426	.500000	.044274	.960000	-.016299
.010000	.037678	.120000	.057233	.550000	.039701	.980000	-.019546
.012500	.039331	.150000	.057906	.600000	.034534	1.000000	-.023286
.015000	.040739						

Table 5.- (Continued)
$\eta = 0$ (lower surface)

ξ	z/c	ξ	z/c	ξ	z/c	ξ	z/c
.000000	.020816	.017500	-.000679	.200000	-.049871	.650000	-.052708
.000625	.016270	.020000	-.001939	.225000	-.053941	.700000	-.045692
.001250	.014435	.025000	-.004277	.250000	-.057370	.750000	-.038317
.002500	.011887	.035000	-.008425	.300000	-.062893	.800000	-.031291
.003750	.009928	.050000	-.013732	.350000	-.066288	.850000	-.025200
.005000	.008318	.070000	-.019957	.400000	-.068211	.900000	-.020931
.006250	.006953	.075000	-.021415	.450000	-.068225	.940000	-.019756
.007500	.005770	.100000	-.028223	.500000	-.066918	.960000	-.020331
.010000	.003778	.120000	-.033239	.550000	-.063623	.980000	-.021869
.012500	.002130	.150000	-.040166	.600000	-.058991	1.000000	-.024907
.015000	.000667						

Table 5.- (Continued)

η = 1 (upper surface)

ξ	z/c	ξ	z/c	ξ	z/c	ξ	z/c
.000000	-.017080	.017500	.007228	.200000	.052401	.650000	.068295
.000625	-.012750	.020000	.008852	.225000	.055222	.700000	.065994
.001250	-.010983	.025000	.011722	.250000	.057708	.750000	.062223
.002500	-.008435	.035000	.016487	.300000	.061831	.800000	.056596
.003750	-.006343	.050000	.022260	.350000	.064994	.850000	.048938
.005000	-.004510	.070000	.028545	.400000	.067322	.900000	.039847
.006250	-.002879	.075000	.029937	.450000	.068904	.940000	.032001
.007500	-.001397	.100000	.036099	.500000	.069824	.960000	.028121
.010000	.001205	.120000	.040244	.550000	.070058	.980000	.024138
.012500	.003456	.150000	.045497	.600000	.069597	1.000000	.019858
.015000	.005446						

Table 5.- (Concluded)

η = 1 (lower surface)

ξ	z/c	ξ	z/c	ξ	z/c	ξ	z/c
.000000	-.017080	.017500	-.036099	.200000	-.054249	.650000	-.018549
.000625	-.021556	.020000	-.037000	.225000	-.054972	.700000	-.008441
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.003750	-.027343	.050000	-.043347	.350000	-.054799	.850000	.016556
.005000	-.028705	.070000	-.045791	.400000	-.052897	.900000	.020562
.006250	-.029856	.075000	-.046320	.450000	-.049588	.940000	.020986
.007500	-.030854	.100000	-.048583	.500000	-.044427	.960000	.020173
.010000	-.032527	.120000	-.050077	.550000	-.037214	.980000	.018431
.012500	-.033905	.150000	-.051993	.600000	-.028364	1.000000	.015755
.015000	-.035080						

Table 6. - Analytical test cases for Rectangular Wing.

Case	M	$Re\times10^{-6}$	α_0	k
1	0.8	3.4	1	0.1
2	0.8	3.4	2	0.1
3*	0.8	3.4	1	0.2
4*	0.8	12.5	1	0.2
5	0.8	3.4	1	0.3
6*	0.8	3.4	1	0.45
7	0.9	3.8	1	0.178
$\alpha_m = 0$, x_α/c = 0.25 and 0.50				

Table 7.- Analytical test cases for RAE Wing A.

Case	M	$Re\times10^{-6}$	h_0/c_r	α_m	α_0	δ_m	δ_0	f	k
1	0.8	3.0	0	0	0.5	0	0	1	0.003
2	0.8	3.0	0	0	0.5	0	0	90	0.26
3	0.8	3.0	0	4	0.5	0	0	90	0.26
4	0.8	3.0	0	0	0	0	1.60	90	0.26
5	0.8	3.0	0	2	0	0	1.60	90	0.26
6	0.9	3.2	0.01	0	0	0	0	90	0.23
7	0.9	3.2	0	0	0.5	0	0	90	0.23
8	0.9	3.2	0	0	0	0	1.76	1	0.003
9*	0.9	3.2	0	0	0	0	1.58	90	0.24
10	0.9	3.2	0	0	0	0	1.63	230	0.60
11*	0.9	3.2	0	1	0	0	1.58	90	0.24
12*	0.9	3.2	0	0	0	3.56	1.58	90	0.24
13*	0.9	3.2	0	0	0	0	3.56	90	0.24

Table 8.- Analytical test cases for NORA Model.

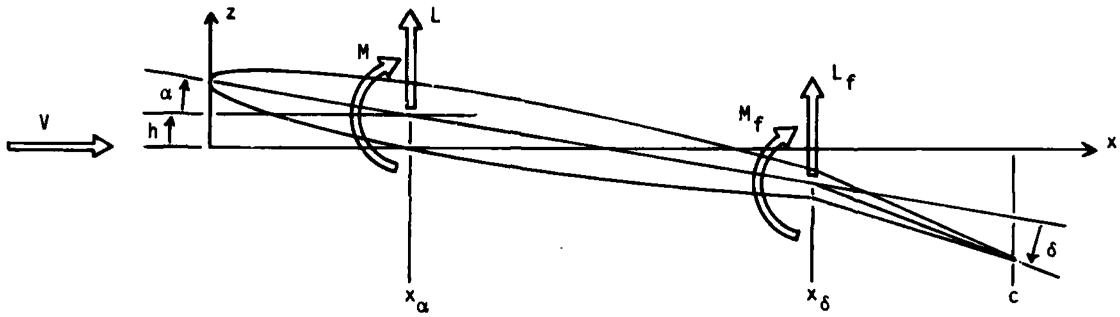
Case	M	$Re\times10^{-6}$	α_m	α_o	f	k
1	0.8	7.8	0	0.5	40	0.31
2*	0.8	7.8	4	0.5	40	0.31
3	0.9	5.5	0	0.5	5	0.035
4	0.9	5.5	0	0.5	40	0.28
5*	0.9	5.5	4	0.5	40	0.28
6*	0.95	5.6	0	0.5	40	0.27
7	0.95	4.6	4.75	0.5	5	0.034
8	0.95	4.6	4.75	0.5	40	0.27
9	1.1	5.8	0.55	0.5	40	0.24

Table 9.- Analytical test cases for ZKP Wing.

Case	M	α_m	δ_m	δ_o	f	k
1	0.30	0	-4.60	0.92	10	0.59
2	0.73	0	0	0.92	20	0.49
3	0.73	0	-5.52	0.92	20	0.49
4*	0.78	0	0	0.92	20	0.46
5*	0.78	0	-5.52	0.92	20	0.46
6*	0.78	2	0	0.92	20	0.46
7	0.83	0	-5.52	0.92	20	0.43
$\alpha_o = 0$						

Table 10.- Analytical test cases for LANN Wing.

Case	M	α_m	α_o	f	k
1	0.72	3	0.5	20	.13
2	0.77	3	0.5	20	.11
3*	0.82	1	0.5	20	.10
4*	0.82	3	0.5	10	.05
5*	0.82	3	0.5	20	.10
6	0.82	3	1.0	10	.05
7	0.82	3	1.0	20	.10
8	0.82	3	0.5	30	.15
9	0.87	1	0.5	20	.09
10	0.87	3	0.5	20	.09
$Re = 8.6 \times 10^6$					



$$q = 1/2\rho V^2 \qquad C_p = \frac{p - p_\infty}{q} \qquad \Delta C_p = C_{p,lower} - C_{p,upper}$$

$$L = qcc_l \qquad L_f = qcc_{l,f}$$

$$M = qc^2c_m \qquad M_f = qc^2c_h$$

$$C_L = \frac{2}{S}\int_0^s cc_l dy \qquad C_M = \frac{2}{Sc_r}\int_0^s c^2c_m dy \qquad C_H = \frac{2}{Sc_r}\int_{\text{control span}} c^2c_h dy$$

$$c_l = \oint_{airfoil} C_p d\xi = \int_0^1 \Delta C_p d\xi$$

$$c_m = \oint_{airfoil} C_p(x_\alpha/c - \xi - z/c \cdot dz/dx)d\xi \cong \int_0^1 \Delta C_p(x_\alpha/c - \xi)d\xi$$

$$c_{l,f} = \oint_{flap} C_p d\xi = \int_{x_\delta/c}^1 \Delta C_p d\xi$$

$$c_h = \oint_{flap} C_p(x_\delta/c - \xi - z/c \cdot dz/dx)d\xi \cong \int_{x_\delta/c}^1 \Delta C_p(x_\delta/c - \xi)d\xi$$

Figure 1. - Wing section and total force and moment definitions.

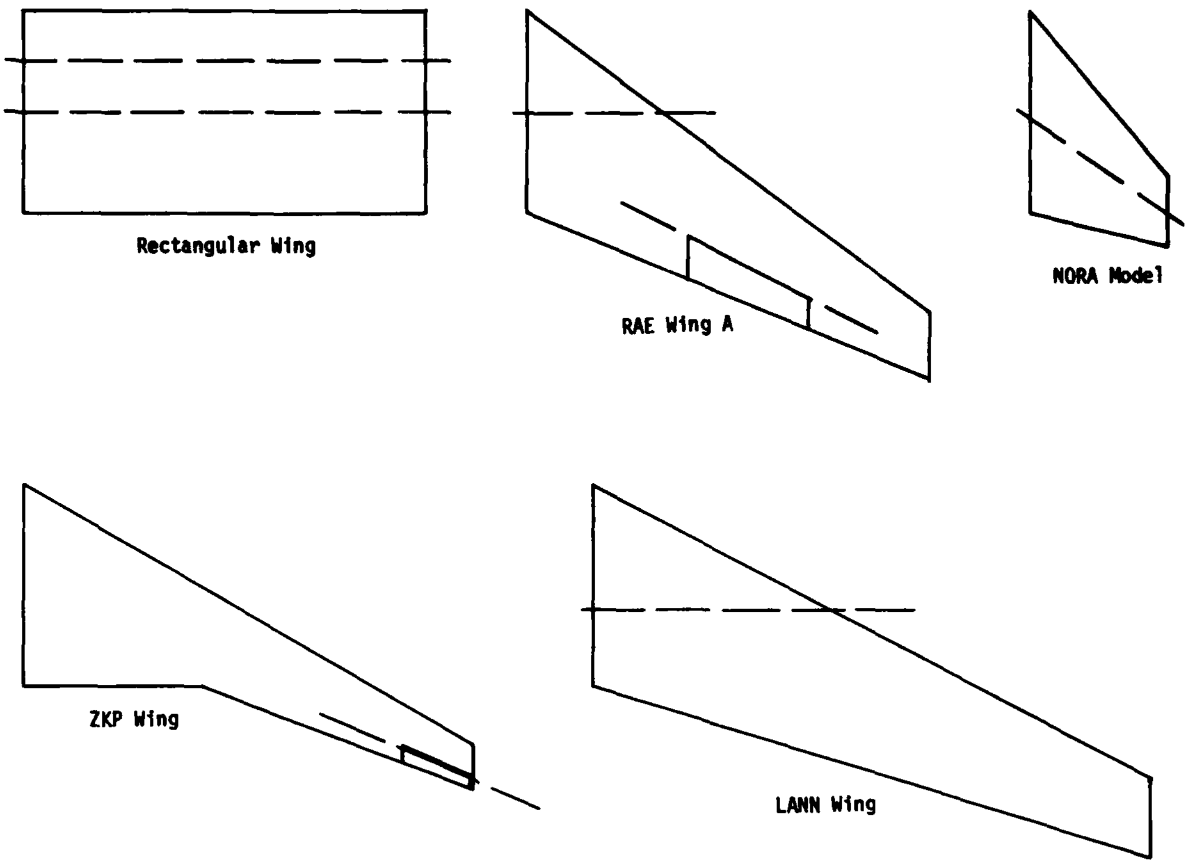


Figure 2.- Sketch of AGARD semispan wings. Broken lines indicate oscillation axes.

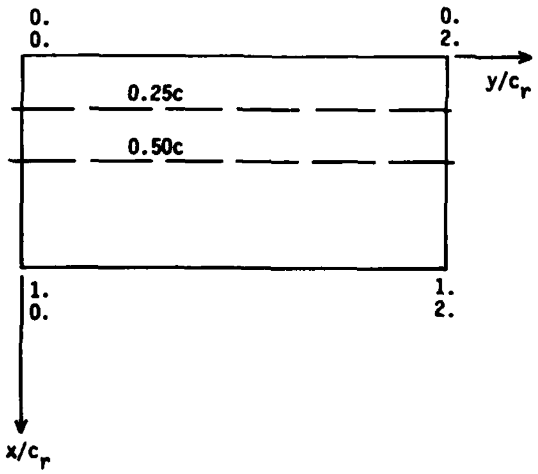


Figure 3.- Rectangular Wing.

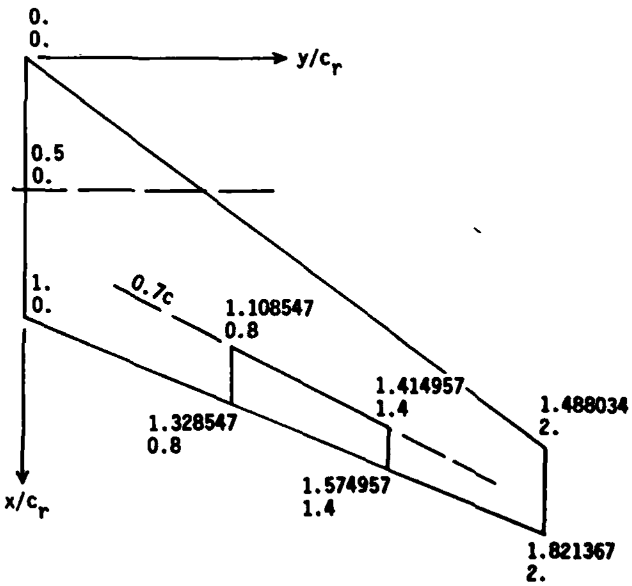


Figure 4.- RAE Wing A.

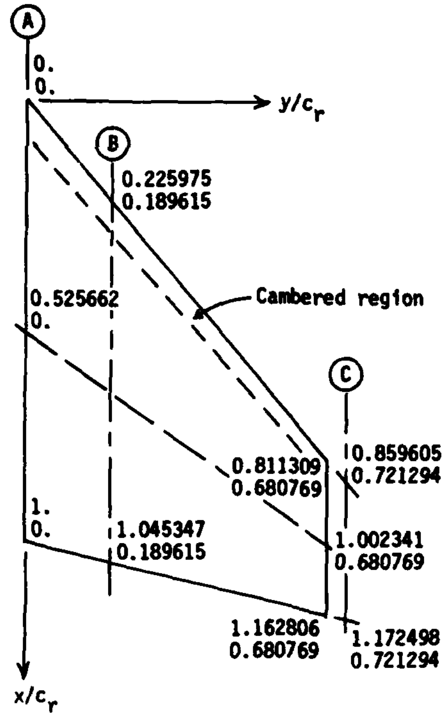
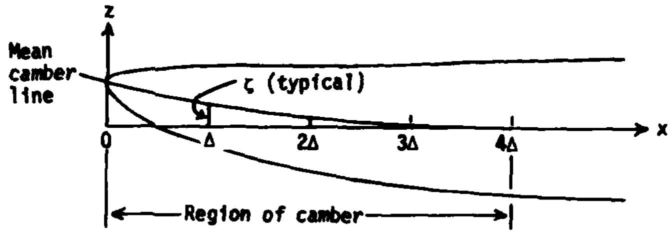


Figure 5.- NORA Model.



x	ζ/c		
	A	B	C
0	.004308	.004566	.006884
Δ	.002442	.002505	.003073
2Δ	.001077	.001065	.000959
3Δ	.000269	.000255	.000123
4Δ	0	0	0
c/c_r	1.000000	.819372	.312892
4Δ/c	.081538	.089395	.159799

Figure 6.- Definition of nose camber for NORA Model.

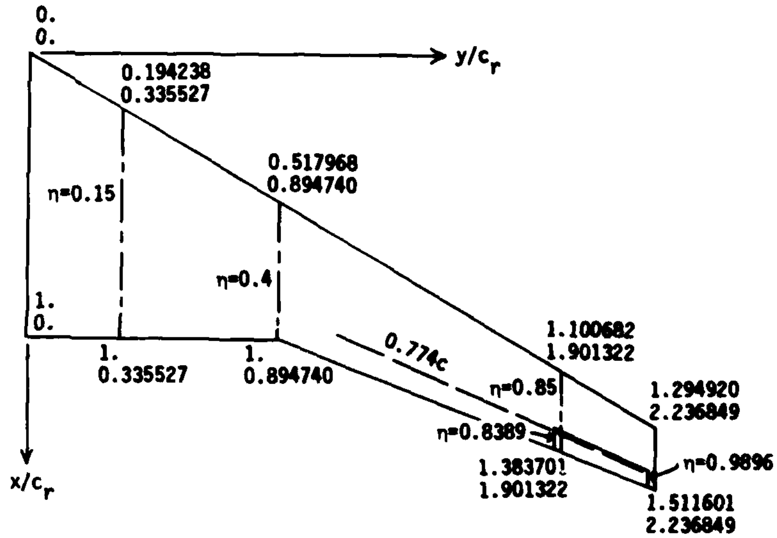


Figure 7.- ZKP Wing.

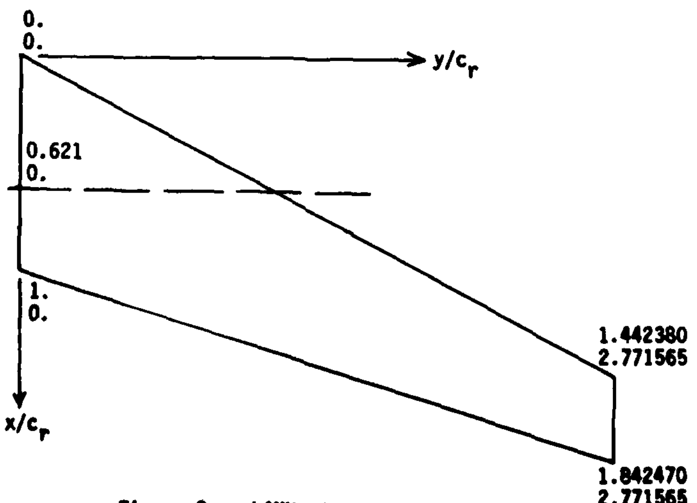


Figure 8.- LANN Wing.

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14. Abstract

The aeroelastician needs reliable and efficient methods for the calculation of unsteady aerodynamic forces in the frequently critical transonic speed regime. The development of such methods may be enhanced by the availability of a limited number of test cases for the comparison of competing methods. This report contains such test cases for five clean, isolated wings. Wing geometric descriptions, airfoil coordinates, and suggested aerodynamic conditions for each are included.

This Advisory Report was sponsored by the Standard Aeroelastic Configurations Working Group of the Structures and Materials Panel.

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